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PROBLEMS IN THE INTERNAL AERODYNAMIC DESIGN OF THE MACH-2 TRANSPORT

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PROBLEMS IN THE INTERNAL

AERODYNAMIC DESIGN

OF THE MACH-2 TRANSPORT*

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SUMMARY

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A survey is presented of the results obtained by the ONERA (Office National d'Etudes et de Recherches Aéronautiques) in the course of preliminary studies of the turbojet intakes and exhausts of the projected Mach-2 supersonic transport plane. Experimental investigations of two-dimensional intakes permit a rational choice to be made between internal and external supersonic compression. Various computational methods employed in the study of the complex flow mixture at the exit are presented and some theoretical and experimental results are compared.

1. INTRODUCTION

In the design of a supersonic aircraft, the air intakes and internal ducts are two essential elements that must be consciously

*This project was reported at the Fifth European Aeronautical Congress held in Venice, Sept. 12-15, 1962.

[redacted]

integrated into the over-all design concept from the very beginning of the project.

In this report we propose to discuss the dimensions and operational control features of these elements for the configuration the French entry in the field is to have. It is already established that it will carry four motors, two beneath each wing, fed by two-dimensional air intakes.

The operating conditions of the engine at different Mach numbers is at the core of the problem; it suggests a principle of simultaneous matching of the intake and exhaust. This principle will be discussed in the first part of the survey.

Subsequently, we will take up the problem of the air intake, which consists in finding the form that affords the best compromise between pressure recovery, or efficiency, and drag. In this field, the ONERA has concentratad its research on a type of inlet that can be controlled by external supersonic compression, with a fairing that presents only a moderate drag; a large-scale test program has now yielded quite satisfactory results. Intakes controlled by internal supersonic compression were also investigated; we shall give experimental data for a comparative appraisal of the two solutions.

The problem of the internal ducts is at least as important as the preceding, and we shall discuss it in the third part of the report.

Some recently developed, and experimentally confirmed, design methods provide a firm basis for selecting optimum solutions at supersonic cruising speeds, but further experiments are required to solve the identical problem in the transonic phase of flight.

2. MATCHING PRINCIPLES

The flow conditions imposed on the entrance and exit may be characterized by two parameters, which are readily derived from a consideration of the data appropriate to the operation of the engine and the law governing intake efficiency. These parameters are the flow factor (mass flow ratio) for the intake and the coefficient of over-expansion (area expansion ratio) of the nozzle. /16

The flow factor of an air intake is the ratio of the engine-required airflow to the maximum flow rate in front of the inlet in the absence of spillage, i.e., the rate of discharge of a stream tube contained in the upstream projection from the edges of the inlet.

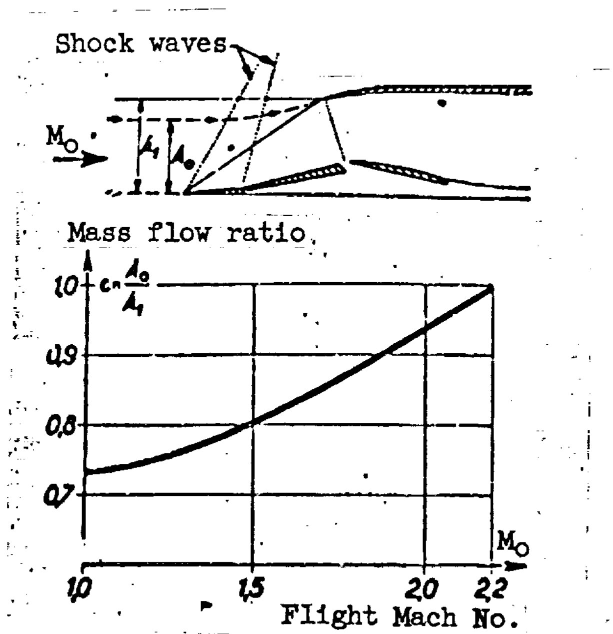


Fig. 1. Airflow adjustment to engine requirement as a function of the Mach number.

Figure 1 plots this factor versus the Mach number for a certain type of motor with given efficiency law that, for the moment, we shall lay aside; the factor is seen to be symbatic with the Mach number, i.e., it decreases as the latter decreases until, in the neighborhood of the transonic mode, it reaches a value of about 0.7. The upstream spillage of the remaining 30% of excess flow is translated into a drag term, called the additive drag, of which we will show the importance and some possible means of mitigation.

The coefficient of overexpansion of the exhaust nozzle is defined in the following way: Let us assume as a first approximation that the exit area A_s of the exhaust has been sized to provide a cruise exit pressure equal to ambient. At lower flight Mach numbers, this condition is satisfied by a section $A^* < A_s$. The coefficient of overexpansion A^*/A_s characterizes the drag occasioned by the pressure drop between A^* and A_s in the flight regime under consideration.

In Figure 2, the low value that this coefficient assumes during transonic flight can be seen.

A calculation of the total parasite drag at flow factors[< 1 is no simple affair, since it is necessary to take into account suction forces on the control lip caused by flow-induced deformation of the leading edges; this effect, by lowering the drag on the fairing amounts to a

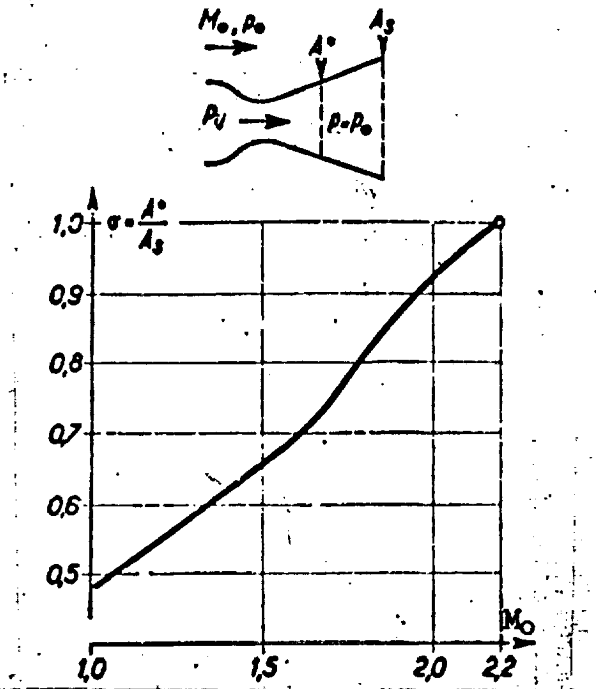


Fig. 2. Nozzle overexpansion ratio.

reduction in the additive drag proper. On the other hand, at the exit there are base drag losses to be reckoned with. These two effects are not at all well defined, especially in the transonic mode; thus, the results of our calculations, shown in Figure 3, give only their orders of magnitude; since these two parasite drags are roughly of the same order, and thus, together constitute almost 20% of engine thrust in transonic mode, it is seen how imperative it is to find a remedy to this situation.

The first idea that suggests itself as a means of avoiding parasite drag at the inlet is to dispose an internal duct around the engine that will bypass the excess air captured by the intake during transonic flight (Fig. 4b). The engine would then operate at its flow maximum without additive drag and then, too, the exhaust nozzle, being better fed, could afford a less severe overexpansion. /17

Such a solution, however, has the inherent drawback that it increases the dimensions of the pod by the space needed to incorporate the bypass, and this increases the cruise drag. In Figure 4c, this disadvantage is circumvented by dumping to the outside the excess air scooped up by the intake by means of a low-angle, low-drag, variable bypass door; the exhaust is then fed by an auxiliary inlet. The loss of charge and the parasite drag involved in this scheme remain to be evaluated.

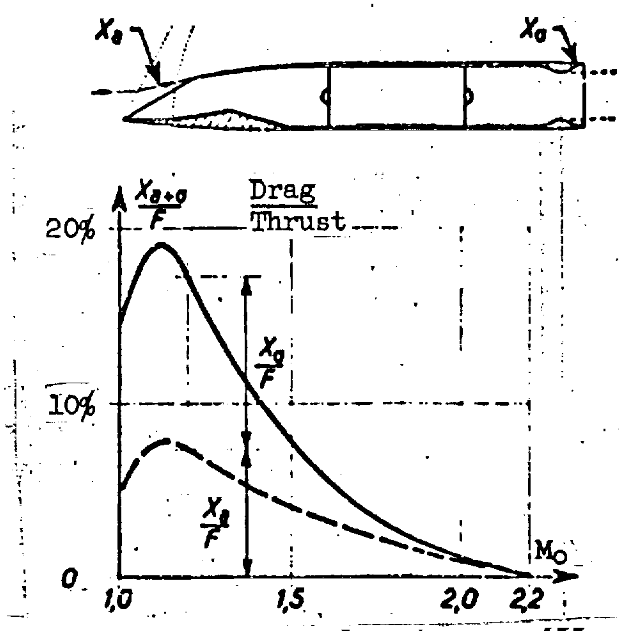


Fig. 3. Parasite drags of upstream spillage X_a plus overexpansion X_σ divided by the thrust, versus the Mach number.

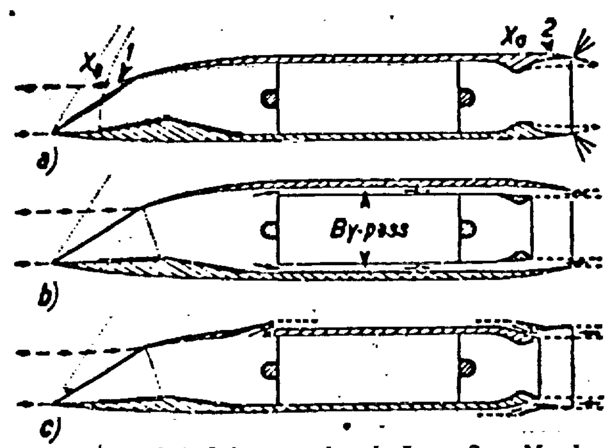


Fig. 4. Matching principles for Mach numbers below cruising.

a) Configuration without bypass. 1. Additive drag.
2. Overexpansion drag

b) Internal bypass matching.

c) External bypass matching with auxiliary intake.

Actually, an air intake can hardly be expected to attain a flow ratio equal to unity in the transonic regime; at best, with good efficiency, it may approach a maximum of 0.85; in this case, the bypass flow is no more than 12 to 15%, but this is to allow reasonable duct dimensions. A compromise might be then to combine an internal bypass with a second additional feed of the nozzle; this injection would consist of the boundary layer removed from the intake and the external boundary layer bled off the auxiliary inlet, the latter removal serving to decrease the capture drag of the supplementary air.

This preliminary discussion suffices for a better understanding of the problems involved in the airflow adjustment of the intake and nozzle. We are now able to approach the problem of the intake in more detail.

3. AIR INTAKE

The conditions that must be satisfied at all Mach numbers are essentially the following: high efficiency, low drag, and large schedule of airflow; the latter implies significant variation of the throat, which in its turn entails a variable geometry.

A two-dimensional intake readily lends itself to these conditions. A diagram of the currently favored configuration is shown in the sketch of the windtunnel setup of Figure 5.

A semituyere mounting was used. A boundary layer bleed at the base of the inlet separates the flow in the manner of an airfoil.

In the prototype version, employing external supersonic compression (ESC), the throat is found at the entrance in a section attached to the cowl lip. An emergent ramp of appropriate shape ensures the focusing of the supersonic shock waves on the lip. This ramp is followed by a fixed wedge (spike) and a second wedge of continuous compression profile. The ramp terminates in the throat to form the upstream edge of the boundary layer bleed located at the foot of the shock wave. (Displacement of the shock pattern in the direction of the nozzle exit provides the transition from the supersonic to the subsonic regime.) Via a hinge coupling, the two wedges together form a variable-angle guide vane, the rotation of which produces a smooth variation of the required compression as a function of the flight Mach number, up to complete blockage of the throat during the transonic phase. The subsonic diffuser wall that follows contains a second articulated vane whose motion is synchronized with the first; the internal boundary layer bleed simply comprises the space between the free ends of the two vanes. /18

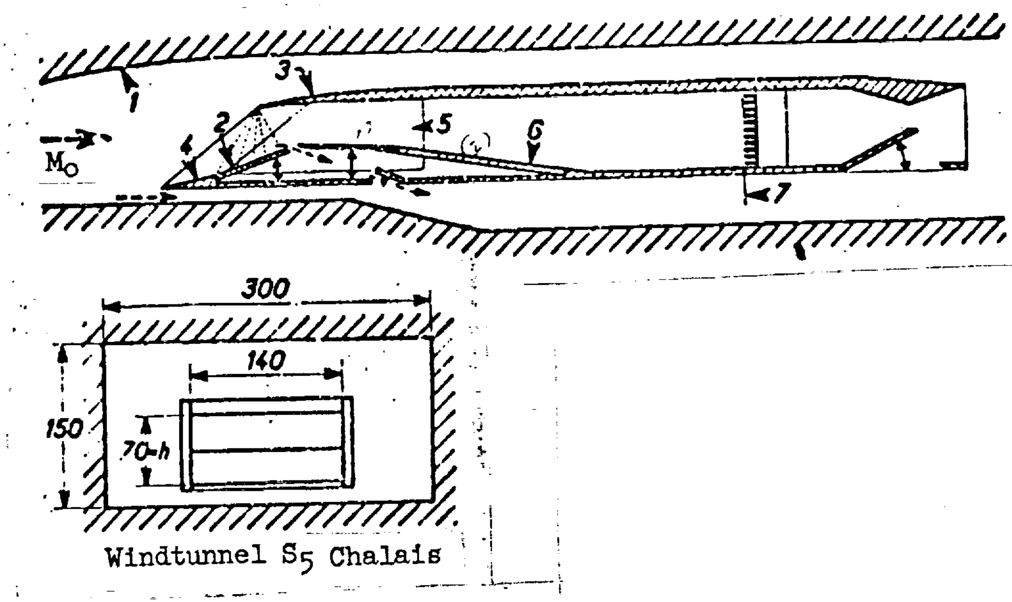


Fig. 5. Two-dimensional air intake.
Test setup in windtunnel. $Re_h = 0.8 \times 10^6$

1. Semituyere
2. Upstream ramp
3. Fairing
4. Nose
5. Window
7. Diffuser vane
8. Vertical axial probe

All the constituent elements of the intake were replaceable in the test model: entrance ramp, upper fairing and lateral leading edges, supersonic compression profile, and subconic diffuser vanes.

Side windows allowed observation of the internal flow. The airflow at the intake and the flow through the boundary layer bleed were controlled by two additional vanes; together they define the cross sections of the sonic throats at the exhausts. Pressure taps at the wall and stagnation point stations provided information concerning the total characteristics and the velocity profile at the end of the diffuser.

The difficulty with this type of external compression intake stems from the sharp deflection of the flow upstream of the entrance (19° at M = 2), since, if a prohibitive nacelle drag penalty is to be avoided, a rapid reverse deflection of the flow must be guaranteed at the very beginning of the subsonic diffuser. Fortunately, the presence of an internal bleed prevents the detachment of the boundary layer that this deflection, in conjunction with the pressure gradient at the foot of the normal shock, would inevitably entail.

The configuration shown in Figure 6 represents one of the better compromises obtained at Mach 2 after numerous trial designs of the shape

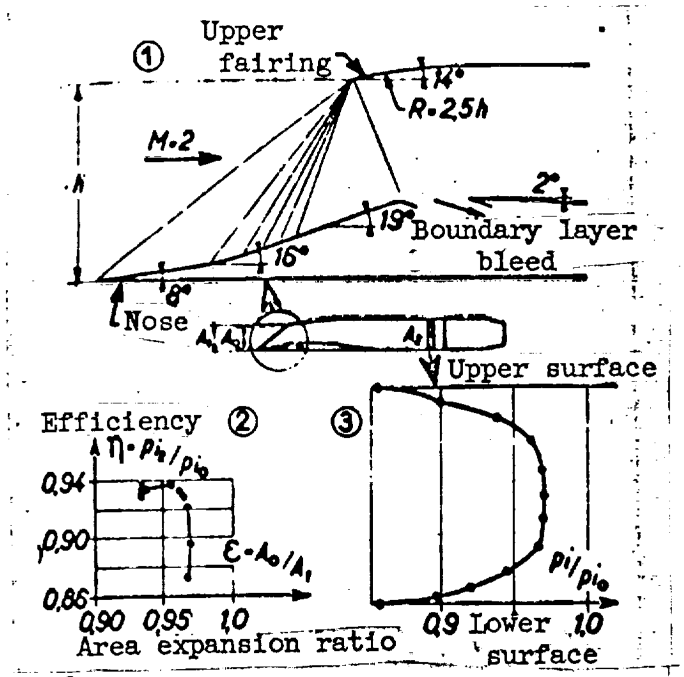


Fig. 6. External supersonic compression at M = 2.

1. Profile definition
2. Total characteristic
3. Axial pressure distribution

and length of the upstream profile of the shape of the external fairing, and of the contour of the boundary layer bleed. Downstream of the bleed, the flow changes from an initial slope of 19° to a negative slope of -2°, resulting in a pronounced curvature, which is compatible with the cowl fairing found to be most favorable from the point of view of external drag.

In spite of this strong curvature, the homogeneity of the flow at the diffuser end is only slightly affected, as is seen from the stagnation pressure distribution in the plane of symmetry at the end of the diffuser and the total characteristics of the flow. The average efficiency of this configuration at Mach 2 is 0.935 in the critical regime, while the boundary layer bleed amounts to only 2.5% of the total flow.

Figure 7 presents the schlieren flow visualizations.

Tests of other forms of supersonic compression profile; using the same fairing, often led to an impasse when attempts were made to adjust the guide vane length upstream toward the throat: either the internal throat "choked up" and there would be a sudden transition from a supercritical regime (diffuser still supersonic in its initial section, normal

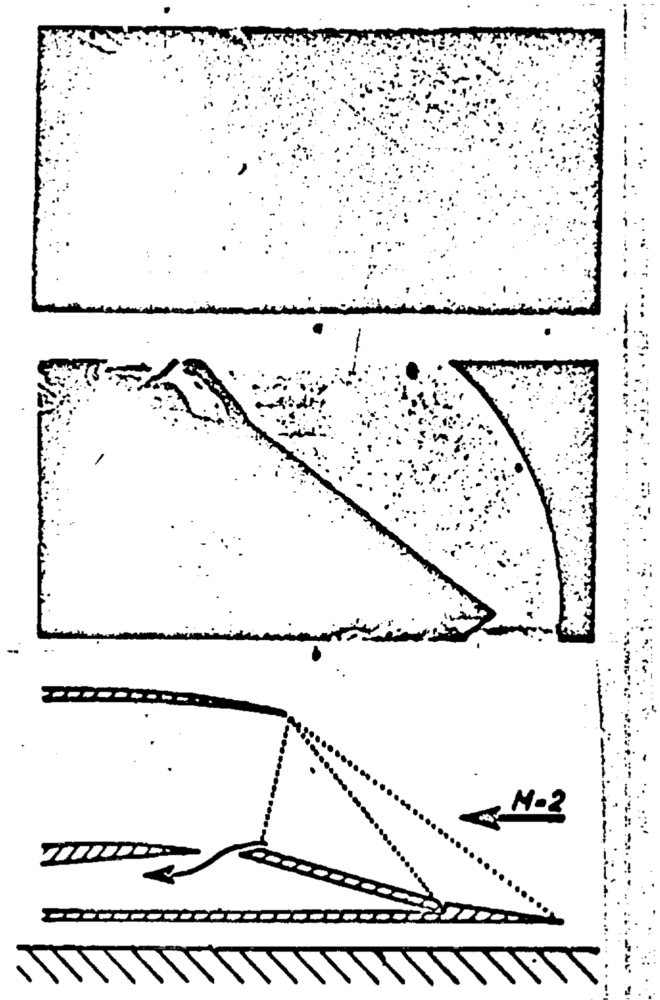


Fig. 7. Two-dimensional intake at Mach 2
Schlieren photographs

A. Boundary layer separation in the absence of a bleed
B. Suppression of separation with internal bleed

internal shock) to a subcritical regime (detached shock in front of the inlet, usually accompanied by duct buzz) or, with reduced guide vane length, there would be a normal shock, properly positioned at the inlet in the critical regime, but with a boundary layer bleed found in an unfortunate position upstream of the normal shock and leading to malfunction. /19

On the other hand, it has been experimentally verified that an intake that is well-matched at Mach 2 retains its optimum characteristics at lower Mach numbers.

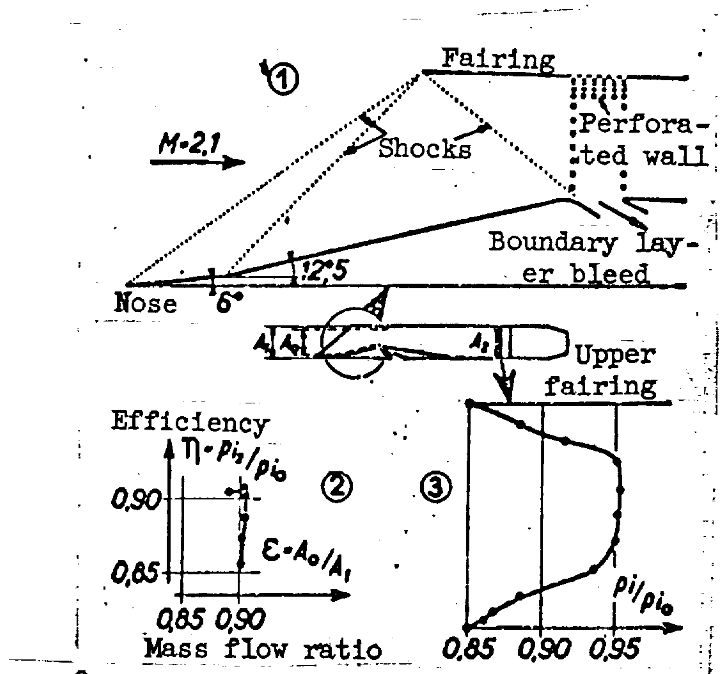


Fig. 8. Internal supersonic compression at M = 2.1

1. Profile definition
2. Total characteristic
3. Axial pressure distribution

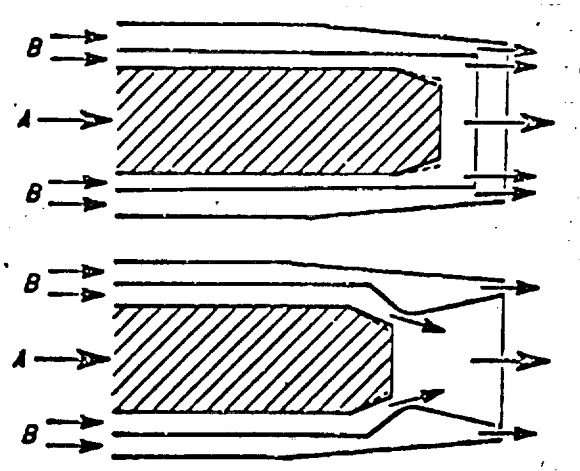


Fig. 9. Airflow adjustment with a variable-area exhaust

A. Main jet B. Auxiliary jet

Figure 8 is a diagram of an intake configuration employing internal supersonic compression (ISC), which we investigated using an

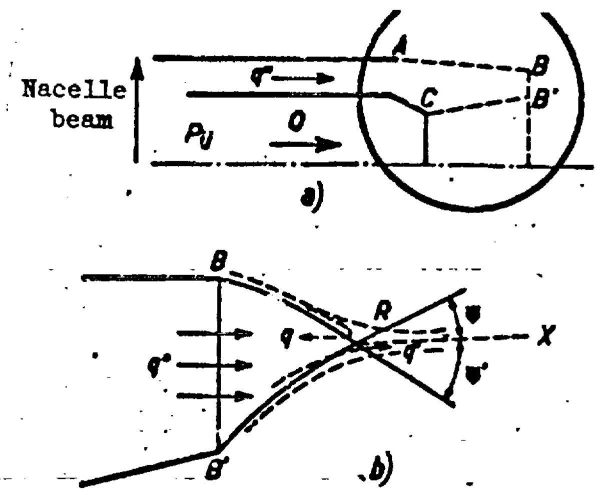


Fig. 10. a) Afterbody study
b) Theoretical flow at the base

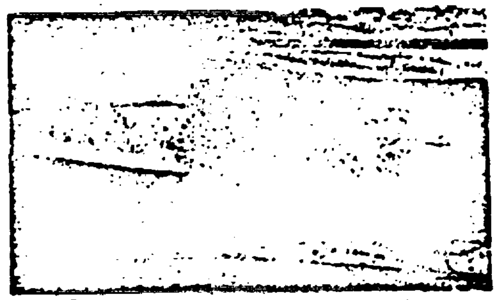
identical test arrangement for comparison purposes. Here, the throat control is inside the inlet, while the internal bleed is secured as before.

This type of intake is characterized by a lower drag, since the initial internal slope of the fairing is zero, but the configuration of maximum efficiency (normal shock directly at the internal throat) is unstable and carries the constant threat of "evacuating" the convergent cone, the shock passing suddenly to a detached position in front of the inlet; manipulation of the guide vanes is then required to replenish the convergent nozzle.

Disregarding the margin of maximum over useful efficiency necessitated by this version, the tested maximum efficiency of an acceptable subsonic diffuser length at the limit of stability is always inferior to the performance of a ESC at Mach 2.

In the same figure are presented the best results attained at M = 2.1; they were bought at the cost of installing boundary layer bleeds at four walls forward of the throat. The maximum efficiency at the "evacuation" limit is 0.905.

The stagnant pressure distribution in the plane of symmetry at the end of the diffuser is less uniform than in the ESC, but it should be noted, however, that the test Mach number is different (2.1 instead of 2).



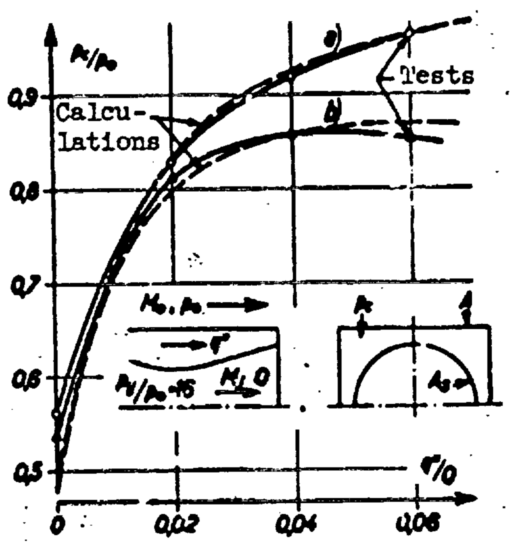


Fig. 11. a) $M_j = 2.24$. $A_s/A = 0.490$.

b) $M_j = 2.55$. $A_s/A = 0.655$.

It should be further noted that the advantage that the ISC is expected to afford in a lower nacelle drag may prove wholly illusory if the cross section of the motor block is wider than the intake.

Finally, considering the need for a more complex boundary layer bleed, the advantage of the ISC is seen to dwindle in the given Mach region.

To conclude this section, and not wishing to pass a definitive judgment on either one of the two solutions, we will only state that our results with external supersonic compression complemented by reduced nacelle drag assure us that there will be continued interest in this type of intake for the Mach 2 class of aircraft.

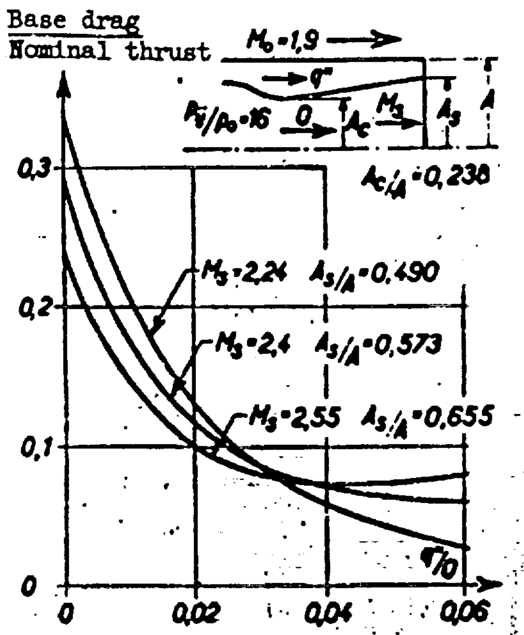


Fig. 12. Effect of injection on the base drag.

4. JET NOZZLE AND AFTERBODY

Still another problem, but a more complex one, is the control of the afterbody of the engine assembly. This involves a study of a convergent-divergent nozzle with a throat area controlled to vary as a function of Mach number and engine schedule by means of an auxiliary injection such as that described in the first portion of this report. /20

Several solutions of the type shown in Figure 9 have appeared in the literature. We cannot consider all of these projected solutions with the same enthusiasm and in the same detail, since numerous pertinent tests remain to be run; but rather, as an example of interest, let us describe the method favored in [1] as a guide to research in afterbody contour optimization, a problem that has proved untractable to the purely empirical approach. We begin with a convergent-divergent nozzle with secondary injection at the throat.

The problem as it appears to the designer is presented schematically in Figure 10.

The maximum cross section of the jet engine (point A) and the throat dimension (point C), as well the flow characteristics of the exhaust gas (flow Q, pressure, etc.) are given: What profile is the afterbody to have to provide maximum thrust? And what benefit may be derived from a certain secondary injection q" in the rear section?

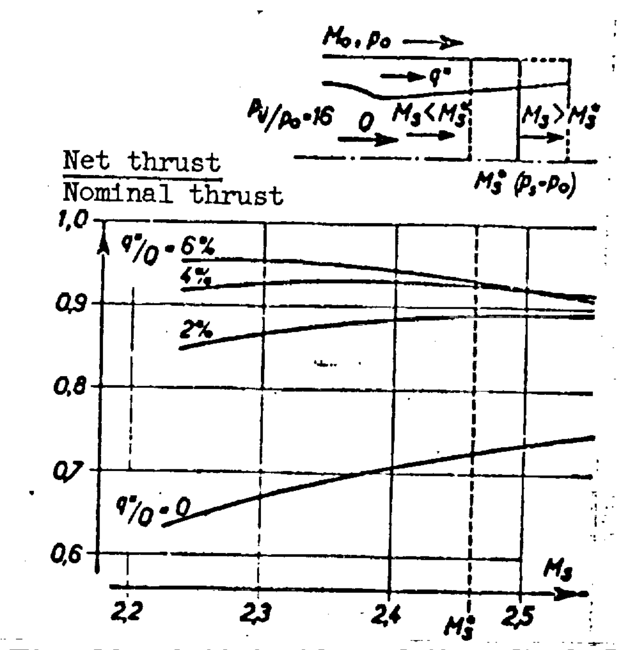


Fig. 13. Optimization of the afterbody.

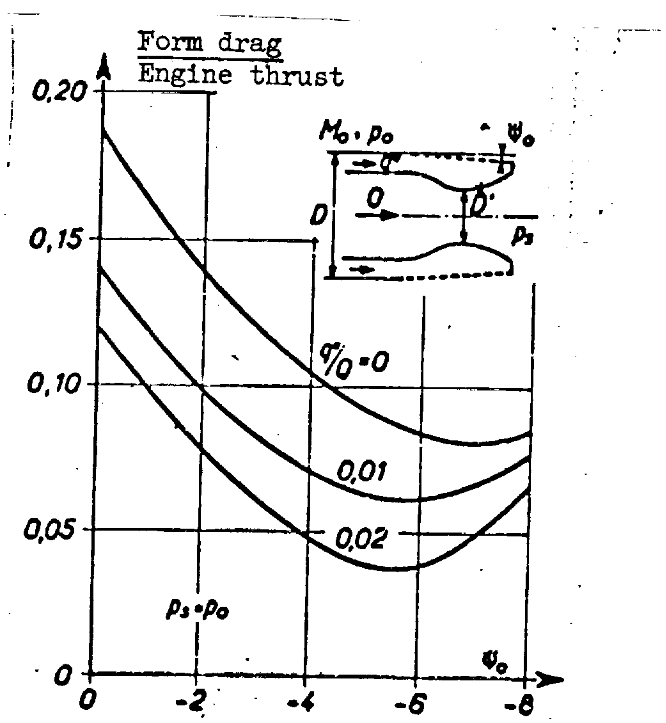


Fig. 14. Effect of shielding on the form drag. $M_0 = 2$.

As parameters characterizing the solution, we have the geometrically defined dimensions of shielding (AB), throat (BB'), and supersonic divergent section (CB'), along with the secondary-to-primary flow ratio q''/Q.

Of course, an optimum solution for one Mach number M_0 is not necessarily optimum for other regimes; the ultimate choice must be some compromise.

We will limit the discussion to the cruise mode near Mach 2.

An evaluation of the forces acting on AB and CB' in the transonic regime presents no special difficulty, so that the search for a key to the problem devolves to a discussion of the mixture of primary and secondary flows across various sections of the throat BB' for a given injection.

The proper approach to the solution of this problem is now well understood. A recent description was contained in the report of P. Carrière [2] on the occasion of the Second Lilienthal Conference.

Let us briefly review the method of attack (Fig. 10).

For a given flight phase, we select an arbitrary value p_1 of the throat pressure; application of the theory of a perfect fluid to the two expansions at B and B' and to the primary-secondary mixing region beyond RX leads to the two stream divergences

$$\Psi = (BR, RX) \qquad \Psi = (B'R, RX),$$

from which can be derived the external airflows q and q'; viscous forces in the stagnant area retard these flows until they are deflected by the back pressure produced at point R in this same region.

Depending on conditions, these flows are positive or negative (in the case of the figure, for example, the flow q' is actually evacuated from the region of stagnation, and should thus be considered negative).

If, furthermore, an auxiliary injection q" is introduced into the stagnant area, the latter receives at the given pressure p_1 a total flow increment equal to the algebraic sum

$$q + q' + q''.$$

It is clear that conservation of mass in the region of stagnation requires that this sum be equal to zero; this condition permits p_1 to be determined by successive approximations.

Of course, actual phenomena are much more complex. Let us only recall two complementary results obtained in [1]:

1. Injection of an auxiliary flow q" is always accompanied by an increase of momentum $(q''V_j)$; these two increments play opposite roles, the first tending to increase the throat pressure while the second decreases it.

The result is as if a pure mass flow (lacking velocity) were injected, equal to $q'' \left(1 - \frac{V_j}{u_1}\right)$, where u_1 is the velocity of the external separated flow and V_j is the injection rate.

Thus, for a given q", it is to great advantage to reduce V_j as far as possible.

2. The boundary layers of the main and secondary flows create slight perturbations at the throat section; there are theoretical means available for their evaluation, but within the scope of an optimum design calculation, such effects can be neglected.

After this hurried review of the principles underlying our design efforts, let us indicate some of the practical results recently obtained by the ONERA. /21

The tests were performed in windtunnel S8 at Chalais-Meudon at Mach 1.9. The mock-up was mounted in the forward blast region on a rectangular beam 54 X 58 mm, with the afterbody unshielded. Two divergent cones of revolution with 13° half-angles were tested; their Mach numbers M_j in the nozzle exit plane were 2.55 and 2.24, respectively.

The auxiliary injection q" at the throat attained 6% of the primary jet.

The pressure generated in the tunnel was p_i = 1 atm; the jet pressure p_{ij} equalled 2.4 atm.

The calculations were made by assuming the external fairing to be a surface of revolution closely fitting the engine contour.

A comparison of theoretical and experimental results is presented in Figure 11 as a plot of throat pressure over ambient pressure p_0 versus the injection ratio.

The existence of a maximum is clearly evidenced, which signals the increasing influence of the momentum imparted by q" as the latter increases while the orifice area is held constant.

Consequently, in Figure 12 we reproduce the dependence of the throat drag on injection ratio for three nozzles of different exit Mach number (nominal engine thrust in the cruise mode is taken for reference).

Thus, to find the total thrust in the rear section, it is sufficient to plot the variation of the divergent cone thrust as a function of the exit Mach number, and then derive the corresponding drag for various values of q"/Q. Thus, for each injection, an optimum exit area emerges, as is seen in Figure 13. /22

This diagram shows, in particular, that the currently fashionable term "matched nozzle", employed to define a nozzle having a pressure in the exit plane equal to ambient, can only confuse matters. Actually, only a family of curves, such as that in Figure 13, can permit a comparison of rear thrust values in diverse regimes and a resultant cone truly matched to the design requirements.

As we noted earlier in more refined studies, there is certain value in seeking an additional means of improving the thrust in the aftersection through some modification of the rear fairing.

On paper, such an approach seems ready-made, since the pressure at the fairing is always greater than the pressure at the throat in the absence of a fairing. In reality, a detailed calculation must be made, since the presence of the fairing alters both the direction and magnitude of the external flow before its separation, and as a result, other things being equal, there is a change in the throat pressure.

Calculations of this type, performed under identical windtunnel conditions, but on a slightly different mock-up, give a good illustration of the possible advantages of a systematic study of the afterbody (Fig. 14).

5. CONCLUSION

Of the multitude of problems connected with the internal aerodynamics of the Mach-2 Supersonic Transport, several still remain to be brought to a satisfactory conclusion: the final choice of the type of intake and the matching of its geometry to the flow field of the wings; the choice of bypass principle; and the selection of the dimensions, control mode, and configuration of the afterbody. At the ONERA, these studies are pursued not only in the cruise regime, but at all the intermediate speeds as well, especially the transonic.

The theoretical and experimental results presented here may give some evidence of what is involved in such research.

Manuscript submitted, Sept., 1962

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